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(56) Documents cited
GB 2097480 A GB 1151937 A GB 0572916 A
EP 0081436 A US 4451203 A

(58) Field of search
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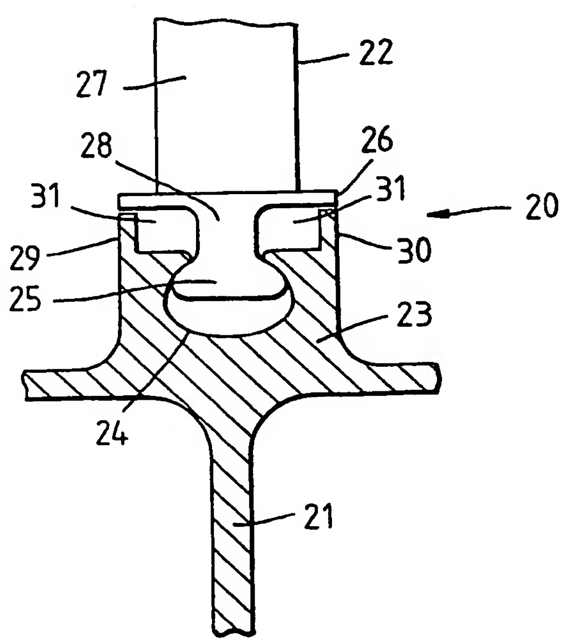
(54) Bladed rotor for a gas turbine engine

(57) A gas turbine engine bladed rotor 20 comprises a rim 23 provided with an annular blade root 25 retention groove 24. Aerofoil blades 22 are mounted in the groove 24. Each blade 22 is provided with a platform 26 which cooperates with flanges 29, 30 which extend radially outwardly of the disc rim 23 to define an enclosed chamber 31. The chamber 31 provides thermal shielding of the disc rim 23 from the gas path over the aerofoil blades 22.

Fig. 2.



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At least one drawing originally filed was informal and the print reproduced here is taken from a later filed formal copy.

Fig.1.

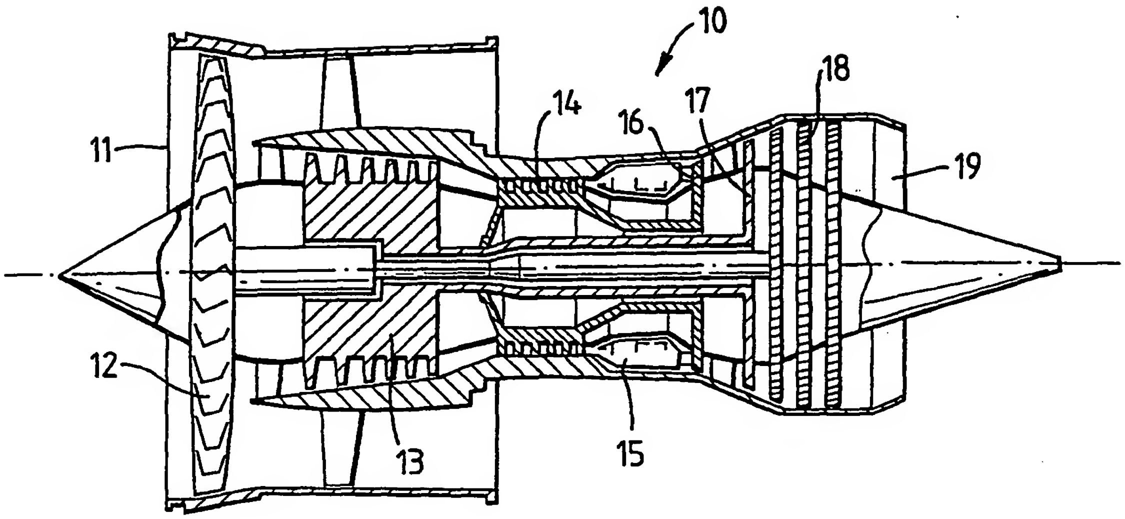
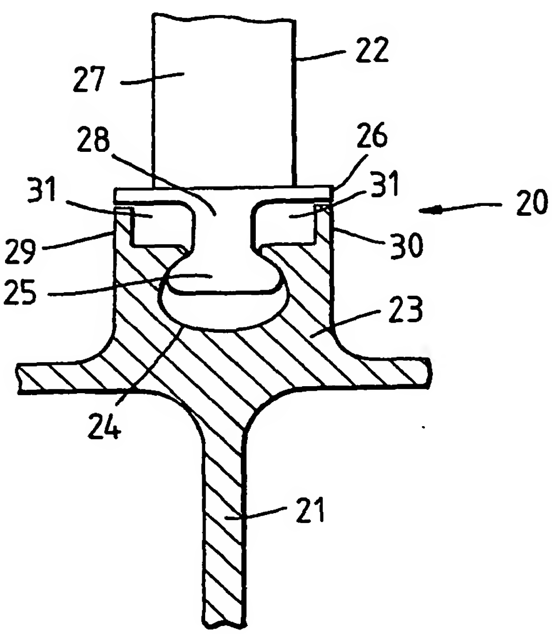


Fig.2.



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BLADED ROTOR FOR A GAS TURBINE ENGINE

This invention relates to a bladed rotor for a gas turbine engine and is particularly concerned with a bladed rotor for a gas turbine engine axial flow compressor.

There is a constant quest to improve the performance of gas turbine engines while at the same time reducing their cost and weight. These objectives are frequently in conflict with each other. One area where this is evident is the axial flow compressors of such engines. In the higher pressure regions of such compressors, the discs which carry the compressor aerofoil blades are desirably manufactured from titanium alloys. Such alloys are light while possessing acceptable strength characteristics.

As the temperatures which are reached in such higher pressure compressor regions increase to provide improved engine performance, serious problems can occur with titanium alloy rotor discs. As the disc heats up during normal engine operation, it can reach temperatures which are beyond those which are normally acceptable for titanium alloys. Under these circumstances, other high temperature resistant alloys, notably nickel base alloys, are used in place of titanium alloys.

One disadvantage of using nickel base alloys in applications of this type arises as a result of the high rates of thermal expansion which these alloys typically have. During certain engine operating conditions, for instance during engine acceleration, high temperature gradients can be created within the discs. Such temperature gradients introduce high compressive stresses in the disc rim. Similarly during engine deceleration, the rim undergoes a reverse stress cycle so that high tensile stresses are introduced into the rim.

The problems associated with the use of nickel base alloy discs are compounded if the compressor aerofoil blades are mounted in a circumferential groove in the

disc rim. In order to load aerofoil blades into the groove, it is necessary to provide an access slot in the disc rim. Such slots constitute stress-raising features during disc rotation; a problem which is exacerbated by the thermal expansion characteristics of the disc.

It is an object of the present invention to provide a bladed rotor in which the disc rim region of the rotor is exposed to a lower level of thermally induced stress.

According to the present invention, a bladed rotor suitable for a gas turbine engine comprises a disc carrying an annular array of radially extending aerofoil blades, each of said aerofoil blades having a root portion, a aerofoil portion, and a platform interposed between said root and aerofoil portion, each of said root portions being radially spaced apart from its corresponding platform and located in a correspondingly shaped annular retention groove provided in the rim of said disc, said platforms defining a portion of the radially inner extent of the operational gas path over said aerofoil portions, axially spaced apart wall means being provided to cooperate with said disc and the axial extents of said platforms to define a substantially enclosed chamber between said platforms and said disc rim so as to provide thermal shielding of said disc rim from said gas path.

The present invention will now be described, by way of example, with reference to the accompanying drawings in which:

Fig 1 is a sectioned side view of a ducted fan gas turbine engine which incorporates a bladed rotor in accordance with the present invention.

Fig 2 is a sectioned side view of the peripheral region of a bladed rotor in accordance with the present invention.

Referring to Fig 1, a ducted fan gas turbine engine generally indicated at 10 is of conventional general configuration. It comprises, in axial flow series, an

air intake 11, a ducted fan 12, intermediate pressure compressor 13, high pressure compressor 14, combustion equipment 15, high intermediate and low pressure turbines 16, 17 and 18 respectively and a propulsion nozzle 19. The engine 10 functions in the conventional manner whereby air entering the air intake 11 is accelerated by the fan 12. The air flow is then divided with the major portion being directed to provide propulsive thrust. The inner portion is directed into the intermediate pressure compressor 13 where it is compressed. It is then further compressed in the high pressure compressor 14 before being directed into the combustion equipment 15. There it is mixed with fuel and the mixture is combusted. The resultant combustion products then expand through, and thereby drive, the high, intermediate and low pressure turbine 16, 17 and 18 before being exhausted through the nozzle 19 to provide additional propulsive thrust.

The fan 12 is driven by the low pressure turbine 18, the intermediate pressure compressor 13 by the intermediate pressure turbine 17 and the high pressure compressor 14 by the high pressure turbine 16, all via interconnecting shafts.

A portion of one of the bladed rotors which are included in the high pressure compressor 14 can be seen more easily if reference is now made to Fig 2. The bladed rotor, generally indicated at 20, comprises a disc 21, only part of the radially outer region of which can be seen, which carries an annular array of aerofoil blades 22.

The disc 21 is provided with rim 23 which contains an annular retention groove 24. The groove 24 is of constant cross-sectional configuration to receive the roots 25 of the aerofoil blades 22. The roots 25 are of similar cross-sectional configuration to that of the groove 24 so that they constrain the aerofoil blades 25 against the centrifugal loading to which they are subject during normal engine operation.

Each aerofoil blade 22 is additionally provided with a platform 26 and an aerofoil portion 27; the platform 26 being interposed between the aerofoil portion 27 and the root 25. The aerofoil portion 27 is of conventional form and functions in the usual manner as it acts upon the air flowing over it. The platform 26 cooperates with the platforms 26 of adjacent aerofoil blades 22 to define the annular radially inner extent of a portion of the axial extent of the air path over the blade aerofoil portions 27.

The platform 26 and root 25 of each aerofoil blade 22 are radially spaced apart from one another by a shank 28. Additionally the disc rim 23 is provided with two radially extending annular flanges 29 and 30 which are axially spaced apart from each other.

The flanges 29 and 30 terminate just short of the axial extents of the blade platforms 26. Thus the platforms 26, the disc rim 23 and the annular flanges 29 and 30 cooperate to define a substantially enclosed chamber 31 which extends around the radially outer extent of the disc rim 23.

The chamber 31 serves to provide a thermal barrier between the radially outer extent of the disc rim 23 and the air path over the aerofoil portion 27. The air which flows over the aerofoil portions 27 is hot, by virtue of the compression which it has undergone in the intermediate pressure compressor 13 and part of the high pressure compressor 14.

The chamber 31 ensures that the heating of the disc rim 23 by the hot air is slowed, thereby in turn reducing stress-inducing thermal changes within the disc rim 23.

It will be appreciated that although the present invention has been described with reference to the flanges 29 and 30 being an integral part of the disc rim 23, other configurations could be employed if so desired. For instance, the flanges 29 and 30 could be constituted

by separate components or alternatively defined by segmental pieces attached to the undersides of the platforms 26.

Although it has not been described in detail, the groove 24 is provided with a suitably configures loading slot to enable the blade roots 25 to be loaded into it.

Claims:-

1. A bladed rotor suitable for a gas turbine engine comprising a disc carrying an annular array of radially extending aerofoil blades, each of said aerofoil blades having a root portion, an aerofoil portion and a platform interposed between said root and aerofoil portions, each of said root portions being radially spaced apart from its corresponding platform and located in a correspondingly shaped annular retention groove provided in the rim of said disc, said platforms defining a portion of the radially inner extent of the operational gas path over said aerofoil portions, axially spaced apart wall means being provided to cooperate with said disc and the axial extents of said platforms to define a substantially enclosed chamber between said platforms and said disc rim so as to provide thermal shielding of said disc rim from said gas path.

2. A bladed rotor as claimed in claim 1 wherein each of said root portions is spaced apart from its corresponding platform by a shank.

3. A bladed rotor as claimed in claim 1 or claim 2 wherein each of said wall means is attached to said disc.

4. A bladed rotor as claimed in claim 3 wherein each of said wall means is integral with said disc.

5. A bladed rotor as claimed in any one preceding claim wherein said rotor is adapted for use in the compressor of a gas turbine engine.

6. A bladed rotor as claimed in claim 5 wherein said rotor is adapted for use in the high pressure compressor of a gas turbine engine.

7. A bladed rotor substantially as hereinbefore described with reference to and as shown in the accompanying drawings.

Patents Act 1977
Examiner's report to the Comptroller under Section 17 (The Search Report)

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Relevant Technical fields

(i) UK Cl (Edition K) F1V (VCM VCN VCP)

(ii) Int CL (Edition 5) F01D F04D

Search Examiner

C B VOSPER

Databases (see over)

(i) UK Patent Office

(ii)

Date of Search

30 JUNE 1992

Documents considered relevant following a search in respect of claims 1 TO 7

Category (see over)	Identity of document and relevant passages		Relevant to claim(s)
X	GB 2097480 A	(ROLLS-) see Figure 2 in particular	1 to 6
X	GB 1151937	(MINISTER) see Figure 3 in particular	1 to 6
X	GB 572916	(ARMSTRONG) see Figure 1	1 to 4
X	EP 0081436	(SOCIETE) see Figure 4	1 to 7
X	US 4451203	(LANGLEY) see Figure 2 in particular	1 to 6

Category	Identity of document and relevant passages	Relevant to claim(s)

Categories of documents

X: Document indicating lack of novelty or of inventive step.

Y: Document indicating lack of inventive step if combined with one or more other documents of the same category.

A: Document indicating technological background and/or state of the art.

P: Document published on or after the declared priority date but before the filing date of the present application.

E: Patent document published on or after, but with priority date earlier than, the filing date of the present application.

&: Member of the same patent family, corresponding document.

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